NMP Flight Project (Project Status)	Technologies	Technology Description	Validation Status	Post-NMP Applications
DS1 (Mission Complete)				Primary technology validation mission ended on 18 September 1999 after encounter with Asteroid 9969 Braile. Extended science mission to rendezvous with Comet 19P/Borrelly successfully completed on 22 September 2001. During the extended mission, the technologies underwent another two years of test and evaluation.
	Autonomous Optical Navigation (AutoNav)	The AutoNav system is a set of software elements that interact with the imaging, attitude-control, and ion propulsion systems aboard DS1. The principal functions of AutoNav were: • Provide critical ephemeris information to other onboard systems, such as the attitude control system • Plan and execute various navigation-related activities such as image taking and processing, ion propulsion system (IPS) thrusting events, and trajectory correction maneuvers (TCMs) • Perform image processing • Perform orbit-determination computations • Perform computations relative to IPS events and TCMs. AutoNav was designed to autonomously determine the position of the spacecraft using images of distant asteroids. AutoNav then computed changes to the spacecraft course using the scheduled IPS thrusting profile (if present) or with discrete TCMs. Validation of the AutoNav system was accomplished through its use as the principle navigation system on the DS1 spacecraft.	Validated (TRL7)	**Deep Impact** Simplified version of the DS1 AutoNav was used on both the Deep Impact impactor and the fly-by spacecraft **Stardust** AutoNav camera pointing software used for Asteroid Braille encounter during DS1 primary (technology validation) mission was modified to locate the nucleus of Comet Borrelly during the extended mission. Subsequently, this modified software was uplinked to the Stardust spacecraft to enable precise location of the nucleus of Comet Wild-2. **Gulliver** Sample return from Mars moon Phobos **AFRL BAA-VS-07-03** Proposed for AFRL Space Situational Awareness study in January, 2007. This proposal offered the DS1 Autonomous Optical Navigation technology with the ST6 ISC to demonstrate a compact package for performing tasks related to space situational awareness. **Asteroid Sample Return** Updated version of AutoNav **Comet Sample Return** Updated version of AutoNav
	Beacon Monitor	Beacon Monitor operations refers to a spacecraft-initiated operations concept and supporting technology components. The supporting technology components are the tone subsystem and the onboard engineering data summarization system. In a typical operations scenario, the spacecraft routinely sends one of four X-band tone messages that indicate how urgent it is to track for telemetry. This tone is received at a smaller aperture antenna than would be required for mission telemetry. If the tone indicated that telemetry was required, a summary of the important telemetry data stored onboard since the last contact would be downlinked via a normal telemetry link. The advantages of using this technology fall into three categories: reducing mission cost, reducing Deep Space Network (DSN) loading, and reducing mission risk.	Validated (TRL7)	**New Horizons** Launched January 19, 2006 on a flyby mission to Pluto and its moon Charon in 2015. This is the first mission to operationally use hibernation in flight and the associated beacon communications mode technology demonstrated by the DS1 mission.
	Ion Propulsion System	The DS1 spacecraft used a single, 30-cm diameter xenon-ion engine, provided by the NASA Solar electric propulsion Technology Readiness (NSTAR) project, as the primary onboard propulsion system. This propulsion system was designed to produce a thrust of 20 mN to 92 mN with a specific impulse of 1950 seconds to 3100 seconds over a input-power range of 500 W to 2300 W. The engine-design life is 8000 hours at the full-power-operating point. The Ion Propulsion System was designed to deliver a total delta V of 4.5 km/s to DS1 while using only 81 kg of xenon propellant.	Validated (TRL7)	**Dawn** Features three NSTAR ion engines on spacecraft for rendezvous with asteroids Ceres and Vesta. **Tempo** Pluto flyby mission **Deep Interior** Jupiter probe **New World Explorer** Asteroid Vesta rendezvous **Odyssey** Comet Kopff rendezvous
	Ka-Band Solid-State Power Amplifier (KAPA)	The KAPA was used to amplify the Ka-band 0.5 mW (rf) downlink signal from the SDST to 2.3 W (rf) with an overall efficiency of 13% and a gain of 36 DB. The 0.66 kg KAPA consists of a three-stage radio frequency (RF) driver module and a three-stage RF output module. Input and output WR28 waveguide isolators are used for low voltage standing wave ratio (VSWR) and output module protection. This unit has successfully demonstrated the highest-power solid state Ka-band amplifier ever used for deep space communications. With future improvements in ground facilities and spacecraft hardware, Ka-band holds a potential four-fold increase in data rate in comparison with X-band.	Validated (TRL7)	
	Low-Power Electronics (LPE)	The Low Power Electronics Experiment was designed to characterize the sub-0.25 micron fully-depleted silicon-on-insulator (FDSOI) baseline process developed at MIT Lincoln Laboratory and verify that the inherent radiation-hardened qualities of the technology that have been examined through ground testing hold true in the space environment. FDSOI offers the advantage of providing high performance (>1 GHz operation) from a sub-2.0 V power supply. The resulting reduction in power consumption (~5 times less power than corresponding 0.25-micron bulk CMOS technology), coupled with the SOI technology's inherent resistance to latchup, make this an attractive choice for the design of integrated circuits used in hardware systems for deep space exploration.	Validated (TRL7)	**Semiconductor process in use by U. S. Navy, Honeywell, and BAE Systems** MIT/LL 0.25 micron fully-depleted CMOS was implemented at U. S. Navy SPAWAR semiconductor fab facility in San Diego, CA. Several modules of this process have also been migrated to Honeywell (Minneapolis) and BAE Systems (Manassas).

	A FDSOI test chip was an integrated circuit consisting of n-channel and p-channel transistors as well as a group of 97-stage ring oscillators. This test chip was integrated into a 6U VME-style test board. The FDSOI test chip was not radiation-hardened. The test board was designed to periodically monitor and record any changes in the basic characteristics of the transistors as well as evaluating changes in switching speed by sampling ring oscillator output frequencies as they were exposed to the space environment. The output of dosimeter and temperature-sensing circuits were sampled and recorded at each step of the test sequence to correlate the effects of temperature variation and total dose radiation.		
Miniature Integrated Camera and Spectrometer (MICAS)	The MICAS is a 12-kg integrated camera spectrometer that includes two panchromatic visible imaging channels (operating between 0.5 micron and 1.0 micron), an ultraviolet imaging spectrometer (operating between 80 nm and 185 nm), and an infrared imaging spectrometer (operating between 1.2 micron and 2.4 micron) along with thermal and electronic controls. All sensors share a single 10-cm diameter telescope with a structure and mirror of thermally stable SiC. There are no moving parts in the MICAS, and the detectors are electronically shuttered. Spacecraft pointing directs individual detectors to the desired targets.	Partial Validation During Flight	**Europa Orbiter** **Pluto Flyby** **Mars Reconnaisance Orbiter** SiC optical elements proposed for Mars Reconnaisance Orbiter
Multifunctional Structures (MFS)	Multifunctional Structures (MFS) is a new approach to electronics packaging, interconnection, and data and power distribution. In particular, the MFS concept involves embedding passive-electronic components within the actual volume of composite materials, new approaches to attaching active-electronic components directly to mechanical surfaces, and using surface areas for mounting sensors and transducers. The MFS design approach uniquely combines 2-D/3-D multichip modules (MCMs) and flex-circuit interconnects, advanced composites (for structures), and thermal management. MFS eliminates the bulky components (chassis, cables, and connectors) of current spacecraft and enables the integration of electronic subsystems, such as data-transmission and power-distribution networks, command and data handling (C&DH) subsystems, and thermal management and load handling. This approach will provide nearly a order-of-magnitude reduction in future spacecraft mass and volume. The baseline MFS design consists of a structural-composite panel that has multi-layer copper/polyimide (Cu/PI) patches bonded to one side, heat-transferring devices embedded, and an outer surface acting as a radiator. Electrical interconnects are designed in the Cu/PI layers, circuitry is implemented in the MCMs, and flex jumpers serve as electrical interconnections for power distribution and data transmission. The thermal management devices embedded in the MFS may include miniature heat pipes and various types of high-conductivity thermal doublers.	Validated (TRL7)	**TechSat 21** **DS2 probes** Used MFS flexible cable technology
Plasma Experiment for Planetary Exploration (PEPE)	The Plasma Experiment for Planetary Exploration (PEPE) is a charged-particle spectrometer capable of measuring and resolving the velocity distribution of ions and electrons and the mass composition of ions that make up a wide variety of plasmas found in the solar system. This instrument was designed to measure solar wind plasma and the plasma populations resulting from solar wind interactions with plasmas associated with the outgassing of asteroids and comets. The new technologies demonstrated in the PEPE instrument include the following: • Novel electron and ion optical systems, including an electrostatically swept field of view and time of flight mass analysis, that significantly reduce overall sensor mass and volume relative to performance, • A compact, high-reliability, high voltage system consisting of eight individual power supplies ranging from +/- 3.6 to +/- 15.0 kV, and • Low-resource, high-performance electronics that perform sub-nanosecond measurements and provide very flexible data acquisition and processing capabilities.	Validated (TRL7)	**Europa Explorer** August, 2005
Power Actuation Switching Module (PASM)	The Power Activation and Switching Module (PASM) is a quad-switch module consisting of a rad-hard switch control ASIC and four standalone semiconductor switch chips. Each of the switch chips provides the capability to switch power, isolate faults, and limit in-rush and fault currents. Each switch chip can switch anywhere from 3 to 40 V at 3 A maximum and, as a result, can be used in switching the primary as well as the conditioned (secondary) power. The PASM also includes trip-time control, di/dt control, and provides remote on/off capability and current and voltage telemetry. The high-density interconnect technology for packaging the chips gives the PASM a four-to-one weight, volume, and footprint advantage over existing products.	Validated (TRL7)	**JIMO** Power Switch slice card containing four PASMs proposed. **Europa Orbiter** Power Switch slice card containing four PASMs proposed
Remote Agent	Remote Agent (RA) is a model-based, reuseable, artificial intelligence (AI) software system that enables goal-based spacecraft commanding and robust fault recovery. RA can operate at different levels of autonomy, allowing ground operators to interact with the spacecraft with immediate commands to the flight software, if needed. However, one of the most unique characteristics of RA, and a main difference with traditional spacecraft commanding, is that ground operators can communicate with RA using "goals" rather than with detailed sequences of timed commands. RA determines a plan of action that achieves those goals and carries out that plan by issuing commands to the spacecraft. Actions are represented with tasks that are decomposed on the fly into more detailed tasks and, eventually, into commands to the underlying software. When discrepancies are detected between the desired state and the actual state, RA detects, interprets, and responds to the anomaly in real time. More serious anomalies can be addressed with longer response times by generating a new plan of action while the spacecraft is kept idle in a safe configuration. When the new plan is generated, the spacecraft is taken out of safe configuration and execution resumes normally.	Partial Validation During Flight	**Derivative proposed by MIT for Messenger** **MIT, JPL and ARC autonomy technology** **EO1** Upgraded version of Livingstone software was uploaded and verified on the EO1.
Small Deep Space Transponder (SDST)	The Small Deep Space Transponder (SDST) was developed as a replacement for the Cassini deep space transponder and supports the radio frequency transmit, receive, and radiometric functions, as did previous transponders. Additionally, the SDST provides a significantly greater functional integration by combining the command detection unit (CDU) and telemetry modulation unit (TMU) in one assembly. The integrated design allows for smaller size, mass, and power consumption of the telecom system compared to previous generations of hardware. The SDST is the first deep space transponder using digital receiver technology. In addition, the SDST is the first Ka-band capable deep space transponder, and provides full support of Ka-band downlink functions, including telemetry modulation and radio metrics (coherent Doppler, ranging, and DOR). A high firmware content was implemented in the SDST digital signal processing module, which was designed to work in X-band deep space, S-band NASA Spaceflight Tracking and Data Network (STDN) facilities, and S-band USAF Space Ground Link System (SGLS) transponders. The high firmware content enables many optional capabilities to be provided with only firmware changes, and allows specific tailoring for each mission.	Validated (TRL7)	**All JPL flight projects (to at least 2007) beyond the moon** The SDST is now a standard product offered by General Dynamics C4 Systems. **Dawn** **Deep Impact** **Mars Odyssey** **MER** **MRO**

				Messenger **Stereo** **Spitzer Space Telescope** **Mars Science Laboratory** Will be used on MSL cruise stage **Mars Phoenix** used on cruise stage
	Solar Concentrator Arrays With Refractive Linear Element (SCARLET)	SCARLET is a concentrator solar array for space applications which uses linear refractive Fresnel lenses to focus sunlight onto spaced rows of multi-junction solar cells to produce up to 2.5 kW of electric power. SCARLET served as the power source for the DS1 spacecraft and its NSTAR electric propulsion system. For a given power level, the SCARLET optical system reduces the required solar cell area by approximately a factor of 7. The decreased cell area can significantly reduce solar array cost while at the same time provide state-of-the-art performance. Other metrics include: • Specific power = 45 W/kg, • Wing dimensions: 264 in. x 64 in. (670.6 cm x 162.6 cm), • Wing mass = 27.7 kg (with tiedowns), • Panel dimensions: 45 in. x 63 in. (114.3 cm x 160 cm), 4 panels per wing, 2 wings on DS1 spacecraft).	Validated (TRL7)	**ELLIPSO** Commercial telecom mission in MEO **STSS** Proposal for USAF non-science mission previously referred to as SBIRS Low. **Stretched Lens Array (SLA)** Advanced solar array based on SCARLET that was proposed to NMP for flight validation on ST6 and subsequently selected to proceed into a flight-validation concept definition study effort. The SLA incorporated significant technology advances over DS1 arrays. **Advanced SLA Technology** Technology advances encompass design approaches to employ Stretched Lens Solar Array (SLSA) technology within future space-based power systems generating large power levels in the range of ~100-200 kW.
DS2 (Failure in Flight)				Two 6.5-kg probes launched piggyback on Mars Polar Lander spacecraft. Probes were lost at Mars in December, 1999.
	Advanced Microcontroller (AMC)	The Advanced Microcontroller (AMC) is a rugged, space-qualified, self-contained microcontroller system with an 80C51 processor, volatile and nonvolatile data storage, and analog interface capability. Its minimal mass, volume and power consumption makes it an excellent option for modest computing needs compared to otherwise more complex, general purpose processors. The AMC only requires external power to operate, and is capable of in situ programming making it versatile and easy to use. The electronic circuits are embedded in plastic using a process known as High Density Interconnect (HDI) to ensure survival during the 30,000 G environment. The AMC has modest A/D and D/A conversion capabilities down to -110° C.	TRL5	
	Entry System	The entry system is an aeroshell that delivers the microprobe from atmospheric entry to Martian surface using a single stage descent. Most of the weight is placed well ahead of the aeroshell center to insure that the probe aligns itself in the descent direction. The aeroshell is attached to the Mars Polar Lander spacecraft cruise ring with a three-legged spider assembly, and it is released by cutting steel cables that hold the aeroshell to the Spider. Inside the shell, the microprobe is held until impact with the help of three face arms bonded to the shell front face. At impact, the face arms break off and the aeroshell shatters. The aeroshell consists of a rear-facing backshell (a segment of a sphere centered at the center of gravity of the entry microprobe) and a forward-facing heat shield (a 45°-sphere cone), both made from 0.8 mm thick silicon carbide (SiC), which has the strength to survive launch but is brittle to shatter upon impact. Silicon carbide was selected as the material for the aeroshells, each of which weighs less than 1.2 kilograms. The aeroshell's heatshield is made of an advanced thermal protection system known as SIRCA-SPLIT (silicon-impregnated, reusable ceramic ablator - secondary polymer layer-impregnated technique). This material is capable of maintaining the probe's internal temperature to within a few degrees of -40° C while the heat shield surface experiences temperatures of up to 2,000° C.	TRL5	
	Flexible Interconnects & Umbilical	Flexible interconnects are strips made of alternating layers of Pyrolux ^{TM} and copper traces. Pyrolux ^{TM} is a trade name for a type of thin polyimide polymer or plastic film. Similar Polyimide films such as Kapton ^{TM} are also used in thermal blankets on spacecraft, while the copper traces are similar to the thin copper paths on a computer or radio circuit board. Flexible interconnects are much lighter, more compact and more flexible than standard wire cables. They can also be easily integrated with spacecraft structure. For DS2, they are used between all electronic subsystems, and for the umbilical which connects the forebody to the aftbody. The design specifications are: • Operate between -120° C and +35° C • Survive a 60,000 G deceleration on impact • Umbilical - be capable of surviving and operating within requirements after a dynamic deployment less than or equal to 200 m/s over a length of 1 m. • Umbilical - Accommodate greater than or equal to forty 0.2 mm traces. Before separation, the umbilical is folded like a firehose in a small rectangular box on the aft end of the forebody. When the penetrator impacts the surface, the umbilical is "layed out" of the forebody as it penetrates the surface. This mode of deployment imparts almost no tensile stresses on the umbilical.	TRL5	
	Low Temperature Battery	The probe's batteries were designed to provide 550 milliamp-hours of power at temperatures as low as -80° C (-112° F) after surviving impact decelerations of 60,000 g. To meet these extreme requirements, the DS2 project developed new, nonrechargable, lithium-thionyl chloride battery cells. To improve low-temperature performance and reduce voltage delays the cells used a lithium tetrachlorogallate salt instead of the more conventional lithium aluminum chloride salt. Each probe contained two batteries composed of four "1/2 D-sized" cells weighing less than 40 grams (1.4 ounce) each. The flight battery lot was shown to operate within a range of 6 to 14 volts after a shelf life of 2.5 years.	TRL5	

Penetrator System - Electronics Packaging	High-g packaging techniques were used for both the mechanical and electronics subsystems. The electrical design features chip-on-board (COB) and three-dimensional high-density-interconnect packaging, encapsulated wire bonds and extensive use of flexible interconnects instead of wires. Assemblies were typically bonded together to minimize potential loose parts and to distribute loads evenly. The aftbody electronics substrates were supported by an electronics shelf, fabricated from high stiffness graphite/epoxy composite, and use COB and high density surface mount technologies. The substrates were Low Temperature Co-fired Ceramic (LTCC) and polyimide-glass with gold for wire bond and passive device pads, and palladium silver for soldered connections. For COB assemblies, cavities were built into the substrates to minimize strain relief for loop wirebonds and provide lateral support for the active devices. The interconnect between the substrates was through polyimide film flex circuitry with surface mount solder lead attachment to the substrates. Discrete wiring from the battery cells is through surface mount soldering joints, the connection between the Telecom electronics and antenna is through a miniature semi-rigid coax cable, and external connections through a miniature connector. The electronics is passivated with Parylene and protected by a non-hermetic cover to provide EMI/EMC shielding.	TRL5	
Penetrator System - Mechanical	The penetrator consists of a forebody and aftbody connected by a flex cable. The forebody has a strong external tubular structure terminated by a hemispherical tungsten nose. The structure shelters two major assemblies, the prism arrangement of three SiC backbones for the forebody electronics, and a science block that houses the forebody science payload (Sample Collection/Water Experiment, Impact Accelerometer, Soil Temperature Sensor), and a cavity to stow the flex umbilical. Cable pass-through for discrete wires and Advanced Microcontroller (AMC) flex from the prism facilitate electrical connection with elements that sit on top of the science block. A side-boring drill for sample collection, driven by the DC-brushed actuator, is nested in the prism interior. The aftbody consists of two major parts: a titanium face and a magnesium case which contain the telecommunication electronics, separation switch, crystal oscillators, batteries, descent accelerometer, sun sensor, and antenna. Prior to impact, the forebody is held inside the aftbody cavity lined with a Teflon sleeve. The electronics rest on a graphite epoxy plate, and are protected by an aluminum cover. The antenna is a 5-inch titanium monopole antenna with 6 steel whiskers 2 inches long, and sits in a dielectric sleeve 31 mm from the aftbody center.	TRL5	
Power Microelectronics Unit (PMEU)	The PMEU controls power conditioning, regulation and switching for DS2 forebody electronics. The unit uses Application-Specific Integrated Circuits (ASICs) in which both digital and analog components are incorporated onto a single chip. Custom CMOS ICs are used since they have excellent low-temperature performance, low-power consumption, and shock tolerance. They also meet the demanding DS2 mass and volume constraints. The unit weighs less than 5 g, occupies 5.5 cc, and requires 0.005 mW. The PMEU has the following design and operational characteristics: • Fits on the front and backside of a prism backbone • Is commanded by the Advanced Microcontroller (AMC) • Supplies 10 power outputs with regulated 3.3 V and/or 5 V, or unregulated power from the batteries • Is capable of powering on/off specified loads • Provides undervoltage protection • Has a regulated full load switching efficiency > 80% and a linear efficiency greater than 40% with an input of 8 Vdc. • Is operable within specifications between -120° C (-184° F) and +35° C • Is capable of surviving and performing after a 30,000 G impact.	TRL5	
Sample Collection/Water Experiment	Each probe is designed to obtain a sample of subsurface soil using a small, ruggedized drill run by an electric motor. When the motor is powered on, a latch is released and the drill shaft extends sideways from the forebody, pulling approximately 100 milligrams (1/250th of an ounce) of soil into a small cup which is then sealed. The sample is then heated, turning any water ice in the soil into water vapor. A small Tunable Diode Laser (TDL) emits infrared radiation through the vapor to a detector; if water vapor is present, it will absorb some of the electromagnetic energy. The laser is the same as the tunable diode lasers flown on meteorology experiments on the Mars Polar Lander, but the path length is much shorter and thus has lower sensitivity. The entire TDL assembly is about the size of a thumbnail. During operation, the water detection experiment requires a peak power of 2 Watts. The presence or absence of water ice at a given depth was to be compared to analyses of soils excavated by the robotic arm on the Mars Polar Lander. One hypothesis is that much of the water that once flowed on Mars' surface is now frozen underground; this experiment was to have helped refine theories of the history and global inventory of Martian water.	TRL5	
Soil Conductivity Experiment	The objective of this experiment was to estimate the thermal conductivity of the Martian soil at depth. The temperature sensors measure how fast the forebody cools down after impact, revealing how quickly heat dissipates in the soil. The measurement represents a new instrument approach that requires far less energy than similar previous experiments on planetary missions, which have used onboard heaters to test the soil. On the DS2 probes two Rosemont platinum resistor high-resolution temperature sensors with the following operating characteristics were mounted on the inner metal surface of the probe forebody, one 1 cm below the tail and one 7.5 cm from the tail end: • Precision: 0.02° C over -70° C to 5° C, and 0.01° C over -120° C to -70° C • Accuracy: 0.5° C over all temperature ranges • Recoverable for temperatures between -120° C and -100° C, and 0° C and +35° C. The temperature sensor system was designed to: • Survive 30,000 g • Operate with an input power of < 55 mW.	TRL5	
Telecommunications System	The final DS2 Telecom design yielded a miniaturized UHF radio system compatible with the Mars Global Surveyor (MGS) Mars Balloon Relay (MR) radio and protocol. This design included an Aftsensor Assembly, one of four originally planned telecom chips, and discrete components and packaged parts (many of which had only become available in the last several years before launch due to advances in cellular phone technology). The final layout spanned almost the entire aftbody electronics motherboard. The telecom system packaging design was based on experience gained from over 3 years of impact testing. This miniature radio-transmitter and receiver system weighs < 50 g, occupies 64 cm², and consumes < 500 mW in receive mode and 2 W in transmit mode. The electronics design satisfies the following performance requirements: • Fixed 8 kbps transmit data rate • PSK transmit modulation format	TRL5	

		• 3 mA in receive mode, 50% efficient at 300 mW in transmit mode (including the antenna) • Survive 60,000 g • Operate between -80° C and 30° C • Include an autonomous emergency transmit mode • Capable of collecting 8 analog signals at 15 bit resolution over a one second sample time, and of incorporating the data in S/C telemetry.		
DS2 Follow-on Experiment (Mission Complete)				The Deep Space 2 Follow-On Experiment successfully validated eleven design principles and guidelines for design of devices intended to withstand the decelerations associated with direct entry impact probes. Results of this experiment are found in JPL document JPL D-29083 which can be obtained by calling the JPL Library Reference Desk.
	Design Principle 01	Materials selection and geometry based upon stiffness requirements to limit deflections	TRL6	
	Design Principle 02	Multiple layers of different compliant surfaces between components and supporting structure act to reflect shock pulse and dampen high frequency components of pulse.	TRL6	
	Design Principle 03	Compressive preloading to prevent tensile loading and to minimize deflections		
	Design Principle 04	Design loads based upon calculated average acceleration times 1.5 to 2.0	TRL6	
	Design Principle 05	Uniformly support all components	TRL6	
	Design Principle 06	Clearances between moving parts minimized; this is an extension of fully supporting all components.	TRL6	
	Design Principle 07	Electronic assemblies and fabrication baselined as "Direct Die Attach"	TRL6	
	Design Principle 08	Treat fluid-filled batteries as pressure vessels and assemble to control deflection of case	TRL6	
	Design Principle 09	Internal wiring of battery strain relieved and supported as fully as possible	TRL6	
	Design Principle 10	Stacked planar electrodes with extended insulating separators to minimize likelihood of shorting to case	TRL6	
	Design Principle 11	Fan fold deployment of umbilical from forebody to have no tension deployment	TRL6	
EO1 (Mission Complete)				Primary (technology validation) mission completed on 20 November 2001. In the baseline mission the EO1 flew 1 minute (450 km) behind Landsat 7. The Extended mission approved by NASA HQ in December, 2001 is chartered to collect and distribute ALI multispectral products in response to Data Acquisition Requests (DARs). Data from extended mission are archived and distributed by USGS Earth Resources Observation Systems (EROS) Data Center (EDC).
	Advanced Land Imager (ALI)	The ALI multispectral (MS) instrument is the primary instrument on the EO-1 spacecraft. The key attributes of the ALI are that it is smaller in both size and weight than the Enhanced Thematic Mapper (ETM+) of Landsat7 by a factor of four and requires less power to operate by a factor of five. The ALI has nine MS bands plus a Panchromatic (pan) band, three more than ETM+, but does not have the thermal band. The ALI has increased sensitivity by a factor varying from four to ten, depending upon the band. The spatial resolution of the MS bands is the same as that of ETM+ (30 m), but is better in the Pan band (10 m versus 15 m). The focal plane of the ALI has been designed to provide a signal-to-noise ratio for each band between four and ten times that of the Landsat ETM+. The focal plane consists of five modules, only one of which is populated with detectors (for cost reasons). When the focal plane is fully populated, the detector arrays will cover an entire 185-km swath on the ground, equivalent to Landsat, operating in a "pushbroom" mode.	Validated (TRL7)	**NPOESS**
	Carbon-Carbon Radiator (CCR)	The EO1 spacecraft uses six 28.62-inch x 28.25-inch passive thermal radiators, each consisting of sandwich panels constructed with aluminum honeycomb cores and facesheets. Five of these radiators use standard aluminum facesheets. The sixth radiator uses carbon-carbon (C-C) material for facesheets. Silver teflon coats the external side (i.e., the side facing space) of all radiator panels. The use of high thermal conductivity fibers in C-C fabrication yields composite materials that have a high stiffness as well as high thermal conductivity. Since C-C density is lower than that of aluminum, significant weight savings may be realized by replacing aluminum facesheets with C-C facesheets. In addition, since C-C is a structural material, it may serve both a load-bearing and thermal management purpose. The primary thermal function of the EO1 CCR was to radiate 27.8 W generated by the power supply electronics (PSE) and 16.3 W (peak power)generated by the LEISA/Atmospheric Corrector (LEISA/AC) electronics boxes. The CCR supported the combined weight (60 lb) of the the PSE and LEISA/AC boxes.	Validated (TRL7)	

	Description	Status	Infusion
Enhanced Formation Flying	The EO1 spacecraft incorporates autonomous navigation capability that allows self-contained sensing, judging, and decision making without ground intervention. The Enhanced Formation Flying (EFF) demonstrated by EO1 with Landsat 7 (LS-7) involved having EO1 maintain a one-minute (~450 km) along track separation to within 6 seconds. EO1 also met the requirement that its ground track in the cross-track direction remain within +/-3 km of LS7's track at the equator. The close separation allowed EO1 to observe the same ground location through the same atmospheric region so that paired scene observations between the two satellites could be made. Two autonomous navigation flight software sets located in the AutoCon™ executive were selected for validation on the EO1 mission. GSFC developed an autonomous formation-flying algorithm that accommodates a general set of orbits for multiple spacecraft. JPL developed a second approach with a more simple control algorithm that focused on missions flying ground track repeat orbits. The JPL approach requires only Global Positioning System (GPS) kinematic navigation solutions for orbit knowledge inputs. The EO1 EFF validation successfully accomplished ten formation-flying maneuvers that included reactionary maneuvers, formation maneuvers, and an inclination maneuver. EFF allows satellites to autonomously react to each others orbit changes quickly and more efficiently. It also allows scientists to obtain unique measurements by combining data from several satellites rather than by flying the full complement of instruments on one costly satellite.	Validated (TRL7)	**SAC-C** JPL algorithm to be uploaded to SAC-C. Algorithm will be used to determine if a orbital correction maneuver is required and which maneuver to perform. This information will be relayed to the ground, and the desired maneuver then commanded from the ground.
Hyperspecral Imager (Hyperion)	The Hyperion is a multispectral "pushbroom" instrument capable of resolving 220 spectral bands and represents a significant improvement in performance over the Landsat ETM+ instrument capable of resolving 10 spectral bands. Each pushbroom image frame captured the spectra from an area 30 m along-track by 7.7 km cross-track. The forward motion of the EO-1 spacecraft created a sequence of frames combined to form a two-dimensional spatial image (with a complete spectral signature for each pixel) with a third dimension of spectral information called a "3-d data cube". The Hyperion has a single telescope and two spectrometers: one visible/near infrared (VNIR) spectrometer and one short-wave infrared (SWIR) spectrometer. The Hyperion instrument consists of three physical units: (1) the Hyperion Sensor Assembly (HAS); (2) the Hyperion Electronics Assembly (HEA); and (3) the Cryocooler Electronics Assembly (CEA). The Hyperion has the following characteristics: • Volume (L x W x H, cm): 39 x 75 x 66 • Mass (kg): 49 • Avg. power (W): 51 • Aperture (cm): 12 Crosstrack FOV(deg): 0.63 • Wavelength range (nm): 400 - 2500 • Spectral resolution (nm): 10 • Spectral bands: 220 • Frame rate: 223.4	Validated (TRL7)	
LEISA Atmospheric Corrector (LAC)	The LEISA (Linear Etalon Imaging Spectral Array) Atmospheric Corrector is a hyperspectral imager providing 256-channel continuous spectra in the wavelength range 0.89 to 1.58 microns. The imager employs a state-of-the-art wedged infrared (IR) filter (a linear variable etalon or LVE) placed very close to a two-dimensional IR detector array to produce a two-dimensional spatial image. The LVE is a wedged dielectric film etalon whose transmission wavelength varies along one dimension to provide its spectral resolution. The LAC was developed to correct high-spatial, low-spectral resolution, multispectral imagery for atmospheric effects. This technology is highly desireable because of its inherent mechanical, electrical, and optical simplicity, its low mass, and its robust nature due to lack of moving parts. The LAC is composed of two modules, the optics module and the electronics module. The optics module contains the lenses, focal planes, and electronics necessary to operate the arrays and to transfer the digitized pixel data to the electronics module. The electronics module contains the command and data interface to the spacecraft, the array timing and bias circuitry, the thermal electric cooler circuitry and the instrument power supply.	Validated (TRL7)	**New Horizons** The EO1 LAC is the precursor of the LEISA sensor on the New Horizons mission to Pluto and the Kuiper Belt. The basic engineering design of LEISA on New Horizons is identical to that of the LEISA on EO1.
Lightweight Flexible Solar Array (LFSA)	The LFSA is a lightweight photovoltaic solar array. The new technology features of this solar array are the use of thin-film copper indium diselinide (CIS) solar cells vapor deposited on a flexible polyimide substrate and the use of shape memory alloy (SMA) for the hinge and deployment mechanism. The LFSA consists of two 5-inch by 18-inch panels. Each panel consists of four 4-inch by 4-inch modules with each module containing 15 CIS solar cells connected in series. The Air-Mass-Zero efficiency achieved for these modules is approximately 2%. The SMA used in the LFSA was the classic binary NiTi (50% Ni, 50% Ti) that has a transition temperature of 70° C. The SMA provides a shockless deployment environment and is safer to handle, integrate, and test than conventional pyro-based deployment mechanisms. A dual flexure array deployment mechanism was developed for the LFSA. In this concept, the SMA strips connecting the two panels together and connecting the LFSA array to the spacecraft are manually buckled and folded into the stowed configuration. Application of heat via internally bonded, flexible nichrome heaters transforms the SMA into a different state and causes the hinge to unfold and deploy. Once the hinge has unfolded, power to the heaters is turned off and the SMA allowed to cool back to its original, but unfolded, state.	Validated (TRL7)	
Pulsed Plasma Thruster (PPT)	The PPT is a small, self-contained electromagnetic propulsion system that uses solid Teflon propellant. It can deliver high specific impulse (650 - 1440 sec), very fine impulse bits (90 - 860 micro N-sec) at low power levels (12 - 70 W) and an estimated total impulse of 460 N-sec. The total mass is 4.95 kg. PPT operation is simple. The main capacitor, connected between the anode and cathode, is initially charged to the desired level and then discharged across the face of a Teflon fuel bar. The discharge of the main capacitor occurs when the spark plug is commanded to fire. A minute amount of charged particles is ablated into the electrode gap when the spark plug is fired. These charged particles provide a conductance path that initiates the main capacitor discharge across the gap. The main capacitor discharge ablates a small amount of Teflon. A small percentage of the ablated Teflon is ionized to plasma. A Lorentz force accelerates the plasma and thus produces thrust. Charged-particle to neutral-particle collisions and pressure forces from resistive heating produces additional acceleration of the neutrally charged, ablated Teflon plasma. To modulate the thrust, the magnitude of the impulse bit is varied. Varying the charge time of the main capacitor changes the magnitude of the impulse bit. The length of time the main capacitor charges directly affects the amount of energy in the capacitor and, consequently, the amount of thrust produced during the discharge.	Validated (TRL7)	**USAF Falconsat 3** Micro-PPT used as a dual-capability thruster by providing attitude control in conjunction with primary propulsion for the spacecraft

	Wideband Advanced Recorder/Processor (WARP)	Spacecraft instruments generate data at ever increasing rates. For instance, the Landsat 7 instrument data rate is 150 Mbps. The EO1, a later generation Earth-observing spacecraft, has an instrument data rate exceeding 500 Mbps. The WARP on EO1 is a very high data rate, solid-state recorder. It is computer-based and provides science and housekeeping data acquisition, storage, and transmission functions. The EO1 flight data system is controlled and monitored through a MIL-STD-1773 Data Bus from an onboard command and data handling (C&DH) unit. When commanded by the C&DH unit, the instruments acquire ground images (scenes) and transfer those scenes at high rates to the WARP. The WARP stores the scenes as files in bulk memory. When in contact with the ground station, the spacecraft automatically transmits the recorded scenes to the ground via an X-band phased array antenna (XPAA) at 105 Mbps or via an S-band omni antenna at 2 Mbps. The following are key specifications for the WARP: • Data Storage: 48 Gbits • Record Rate: > 1 Gbps (burst), 900 Mbps (continuous) • Playback Rate: 105 Mbps with built-in X-band RF exciter • Size: 25 x 39 x 37 cm • Mass: 18 kg • Power: 45 W orbit average, 100 W peak	Validated (TRL7)	
	X Band Phased Array Antenna (XPAA)	The XPAA was used to transmit EO1 imaging data stored in the Wideband Advanced Recorder Processor (WARP) to ground. The XPAA operates at a frequency of 8.225 GHz, transmits data at 105 Mbps with a minimum effective isotropic radiated power (EIRP) of 22 dBW, has an integral controller and power conditioner, and communicates with the spacecraft over a MIL-STD-1773 fiber-optic data bus. The antenna aperture consists of an 8 x 8 array of modules, each comprising a dielectrically-loaded circular waveguide, two orthogonal antenna feeds, a phase shifter, and a dual power amplifier. The 64 modules, each approximately 0.7 inches in diameter and 0.5 inches long, are mounted on a printed wiring board, which distributes radio frequency (RF) excitation, logic control signals, and power to each module. The modules are arranged in an equilateral triangle lattice with the module spacing selected to prevent the onset of grating lobes at the maximum scan angle of 60 degrees. The XPAA and support electronics are contained in a single 12 x 13 x 2.9-inch enclosure with a total mass of 5.5 kg. A multiple-layer, wide-angle, impedance matching radome is incorporated into the aperture design to provide a nearly ideal cosine scan behavior at 60 degrees of scan at all scan angles. Radiation from the antenna is left-hand circularly polarized.	Validated (TRL7)	
EO3 Imaging Fourier Spectrometer (Flight Project Cancelled)					Ground validation of sensor assembly (Engingeering Demonstration Unit) was completed at Utah State University Space Dynamics Lab in September 2006.
	Large-Area Detector Arrays	Fast frame rate, large-area IR detector array sensors are well suited for surveillance imaging and Fourier transform spectrometry. Two versions exist, one for very long wavelength coverage from 8.8 to 14.6 um and the other for the midwave spanning 4.4 6o 6.1 um. The format for both versions is an 128 x 128 array of HgCdTe photo-diodes on 60 um centers with frame rates up to to 8 KHz. Key Capabilities: • Format -- 128 x 128, • Pitch -- 60 um, • Frame Rates -- <8 K frames/sec, • LWIR operating temperatre -- 60 K, • Well capacity (selectable) -- 20 M electrons and 100 M electrons, • LWIR Responsivity (8 KHz) -- 30 nV/photon, • MWIR Responsivity (2 KHz) -- 140 nV/photon, • LWIR Noise Equiv. Input -- 1.9e12 ph/(sec-cm2), • MWIR Noise Equiv. Input -- 1.0e11 ph/(sec-cm2), • LWIR Operability (2 x Mean NEI) -- 92%, • MWIR Operability (2 x Mean NEI) -- >99%, • Ionizing Radiation Tolerance -- >25 Krads (Si).	TRL6	
	Lightweight Optics and Structures	The EO3/GIFTS instrument incorporated a silicon carbide (SiC) scan mirror and an all-SiC telescope (mirrors plus metering structure). The silicon carbide approach provides several key benefits: • Wavefront stability in the presence of direct solar loading on the mirrors due to the excellent thermal properties of SiC • Very low mass due to the excellent mechanical properties of SiC • Near-net shape casting of the mirrors in light weight ribbed geometries • Visible-level athermal performance to cryogenic temperatures due to the matched thermal expansion coefficient of the mirrors and metering structure • No need for a focus mechanism due to the athermal performance and the absence of moisture expansion in SiC Optical performance, mass, and thermal performance were verified during ground testing and calibration.	TRL6	**New Horizons** Silicon carbide telescope (mirrors and structure) operating at cryogenic temperatures and providing excellent images. **GOES-R/ABI** Silicon carbide scan mirrors for stable wavefront performance in the presence of direct solar illumination. **DoD Application, airborne reconnaissance** Silicon carbide optics and telescopes for visible spectrum, light-weight, stiff, thermally stable performance in manned and unmanned aerial vehicles. **DoD Application, laser communications** Silicon carbide optical systems are the material of choice for weight-critical lasercom terminals with stressing thermal and solar operating environments.

	Minicryocooler	The Mini cryocooler is a 2-stage, pulse tube cryocooler designed to cool an instrument to 55 K and its associated optics to 140 K. • The cryocooler simultaneously provides 1.5 W cooling at 55 K and 8 W cooling at 140 K while rejecting heat at 303 K • The cryocooler requires 205 W system bus power • The electronic controller provides settable 2nd stage temperature, vibration reduction, and current ripple suppression • The system mass is 10 kg (7 kg cryocooler + 3 kg controller) • The cryocooler is designed for a operational lifetime of greater than 7 years.	TRL6	**DOD national security satellite** This cryocooler was one of several tested for use on a DOD satellite. After successful testing of the cooler for the DOD system requirements, the program was shut down.
	Rad-hard, 14-bit A/D Converter	The Raytheon AL2 is a low-power, 10 megasample/sec, 14-bit analog-to-digital converter (ADC) implemented in CMOS technology. The device, which is radhard to at least 100 Krad total dose, uses a pipelined architecture with an internal sample and hold. The AL2 may be powered using a single +5 V supply. However, separate analog and digital power supplies are recommended. Power dissipation can be optimized to the sampling rate with the external bias setting resistor. +3.3 V or +5 V CMOS digital input/output is supported. The analog input is true differential, but can accept single-ended inputs. Analog input voltage gains of 1.33, 1.67, 2.00, and 2.67 are externally selectable by the user. User supplied external voltage references set the analog input range, the bias current setting, and offset adjustment if desired. User controls also include a low power standby, three-state output buffer and a reset function for synchronization of the output data to a system. Data outputs are in a straight binary format.	TRL6	
ST5 (Mission Complete)				The ST5 mission validated the measurement strategy of micro-satellite constellations to synthesize higher order physical quantities from the collective measurements made by the individual spacecraft through the coordinated collection and analysis of magnetometer measurements taken by three micro-spacecraft flying in formation over the Earth's auroral ovals in low-Earth orbit.
	CMOS Ultra-Low Power Radiation Tolerant (CULPRiT) Logic	CULPRiT is a Complementary Metal Oxide Semiconductor (CMOS) radiation tolerant technology that allows circuits to operate at low voltage. The technology is capable of significant power reduction over current technology, while achieving radiation and latch-up tolerance. For ST5 the CULPRiT chip was used as a Reed Solomon Encoder with the following features: • Core Operating Voltage: 0.5 V • On-Chip 3.3 Vdc Level Shifters • Externally supplied back bias voltage • Radiation Tolerance: Total Ionizing Dose > 100 kRad • Single Event Upset: LET_{th} ~ 20 MeVcm²/mg • No latch up: LET > 60 MEVcm²/mg CULPRiT performed flawlessly over the 90-day ST5 mission, transmitting over 330 million telemetry frames through the Reed-Solomon Encoder without a single bit error, no positive and negative bias voltage instability, or current increases related to radiation. In comparison DRAM on the CD&H observed over 10,000 single event effects in the same radiation environment. CULPRiT powerup was nominal on all three spacecraft. The 0.5 V core voltage remained stable throughout the mission. A slight increase in the 0.5 V current correlated with the gradual increase in temperature over the life of the mission.	Validated (TRL7)	
	Cold-Gas Microthruster	A single Cold Gas Micro-Thruster (CGMT) using nitrogen as propellant provided fine attitude adjustment for each of the three ST5 micro-satellites. The CGMT is part of a welded propulsion system consisting of a propellant tank, pressure transducer, filter, fill and drain valve, associated braces and tubing, and is controlled by thruster control electronics. This thruster can be operated in both pulse and continuous fire modes to achieve both delta-V and attitude control. The latching solenoid valve design provides an order of magnitude reduction in power consumption compared to thrusters based on continuous duty solenoid valves. The mass of the CGMT is 78 g including lead wires. The on-orbit performance of the ST5 CGMT met all performance expectations and showed excellent correlation to ground test data.	Validated (TRL7)	
	Constellation Operations	The three ST5 micro-satellites operated successfully as a constellation for the nominal 90-day mission. The mission demonstrated model-based operations automation and ground communications strategies requiring minimal ground support (including lights-out operations). The GSFC Mission Services Evolution Center (GMSEC) provided a scalable, extensible, ground-system platform that allowed autonomous ground system and spacecraft constellation operations. The GMSEC connected to: • A planning and scheduling system to support ground, real-time, and stored spacecraft commanding; • A model-based operations system that utilizes models developed and refined to automate scheduling and asset re-tasking; • Absolute time-command sequences, relative time-command sequences, and telemetry status monitors to provide spacecraft on-board automation. The ground data system was implemented as a spacecraft independent dual string, prime and redundant, system supporting the ST5 constellation. The ST5 spacecraft incorporate on-board automation through the use of absolute and relative time-command sequences, and telemetry status monitors. The spacecrafts monitor themselves and perform automated failure detection and recovery actions. They are capable of autonomously acquiring the Sun when commanded by the ground or if on-board fault conditions (based on Sun angle) exist.	Validated (TRL7)	**SDO, GLAST, LRO** Use GSFC Mission Services Evolution Center (GMSEC)architecture. **Advance Information System Technology (AIST) 3-yr. award for "An Inter-operable Sensor Architecture to Facilitate Webs in Pursuit of GEOSS** Key topic: Interoperability and demonstration of service oriented architecture for space missions and sensor webs. **Advanced Information System Technology (AIST) 3-yr. award to "Use Intelligent Agents for Form a Sensor Web for Autonomous Mission Operations"** Key topic: Extend effort ...... in which ST5 components turned into mobile agents for use onboard spacecraft with GMSEC/CFS.

ElectroStatic Switched Variable Emittance Coating	The temperature of a spacecraft is determined by the amount of heat generated within the spacecraft and by the emittances of the external surfaces of the spacecraft irradiated by the sun and exposed to the cold, deep-space environment. A variable emittance coating is a thermal control coating that can be switched from a low-emittance value to a high-emittance value by the spacecraft command and data handling (C&DH) subsystem. In this example a thin film of metallized polymer operates as a thermal switch to change between radiative or conductive heat transport, thereby effectively varying the emissivity of the surface. The thermal switching is controlled via electrostatic forces applied from a good thermal-contact voltage. When attracted, the film makes a good thermal contact to the substrate. In this state, the emitting surface of the radiator (i.e., that side of the film facing space) is at the substrate temperature and is in the "high emittance" state. In the released ("low emittance") state, the film separates from the substrate, and thermal radiation becomes the primary heat transfer mode between the substrate and the film. In this released state, the temperature of the film will decrease and radiate to space at this lower temperature. The electrostatically-switched VEC radiator is composed of four individually controlled, equal surface area segments.	TRL6	
Evolved X-band Antenna	Each ST5 spacecraft has two X-band antennas, one each on the top and bottom decks to give approximately a 4pi steradian coverage. The antennas radiate in the Earth Exploration Services (EES) downlink band at 8470 MHz and receive in the uplink band at 7209.125 MHz. These antennas are required to support the miniature transponder. Each is enclosed in a similar radome and are coupled equally using a 3 db power divider. Functionally the antennas are similar, but their designs differ. The South facing antenna is a quadrifilar helix (QFH). The North facing antenna is an "evolved" antenna, designed with software to develope a radiator that meets the required pattern and matching impedance. The computer program builds a wire form radiator. The evolved antenna maximizes the gain especially towards zenith to accommodate the ground network McMurdo station that has a lower link margin in both uplink and downlink. This antenna weighs 217 g and has an RF input power rating of 10 W at transmit with > 85 % efficiency. The antennas on all three ST5 spacecraft performed well during the 90-day mission. Once inaccuracies of the initial orbit vector were resolved and operators were able to point the MCMurdo station antenna correctly at the beginning of a pass, the ground station auto track locked up automatically at the beginning of the pass.	Validated (TRL7)	
Low-Voltage Power Subsystem Components	The ST5 mission used a Low-Voltage Power Subsystem with a low weight Lithium-Ion battery and high efficiency triple-junction solar cells. The Li-Ion battery represents a three-fold improvement in energy density over existing Ni-cad batteries. The ST5 battery consists of six parallel strings of two 1.5 V Li-Ion cells each. It contains internal over-charge, over-temperature, and over-pressure protection. The battery weight is 645 g, has a 8.4 Vdc voltage output (max.) and a 7.5 A-h effective capacity (to 6 Vdc discharge). The Li-Ion battery has a longer life than Ni-Cad batteries and does not remember discharge cycles prior to recharging. The solar array is constructed of a composite facesheet substrate, triple-junction cells, cover glass and associated wiring and interconnects. The ST5 mission carried 8 solar array panels, each with 15 solar cells arranged in 3 strings (5 cells). The cell efficiency is 28.5% (minimum average at 1 Sun intensity). The beginning of life electrical output per panel is (30° C, 1 Sun air-mass zero); - load voltage 10.2 Vdc, current 1.16 A. The performance of the low-voltage power system on the three ST5 spacecraft was better than predicted.	Validated (TRL7)	**Solar Dynamics Observatory (SDO), Time History of Events and Macroscale Interaction During Substorms (THEMIS) , Lunar Reconnaisance Orbiter (LRO)** Li-Ion battery **MESSENGER** Triple Junction Solar Cells
Magnetometer Deployment Boom	The ST5 magnetometer deployment boom is constructed as an assembly of graphite composite tube sections with three segments of beryllium copper "carpenter tape" hinges. The boom weight is 225 g and length 80 cm. The design minimizes magnetic interference with the magnetometer and provides thermal and dynamic stability. For launch the boom is stowed around three sides of the octagonal spacecraft and held in place with a low-shock shape memory alloy activated pin puller. The strain energy developed as the boom is stowed is used to deploy the boom in orbit by ground command. The 3-axis magnetometer is held about 1.5 spacecraft diameters from boom mounting surface to reduce the effects of stray magnetic fields from within the spacecraft body. The magnetometer boom deployed nominally on all three ST5 spacecraft, and magnetometer data have been very stable. By the end of the three-month mission there had been no indications of fluctuations in the magnetometers position with respect to the spacecraft.	Validated (TRL7)	**Geospace Electrodynamic Connections (GEC)**
MEMS Shutter Variable Emittance Coating	The temperature of a spacecraft is determined by the amount of heat generated within the spacecraft and by the emittances of the external surfaces irradiated by the sun and/or see the cold, deep-space environment. The current state of the art is to use appropriate coatings with fixed emittance values. A variable-emittance coating is a thermal control coating that can be switched from a low-emittance value to a high-emittance value by the spacecraft command and data handling (C&DH) subsystem. In this example the radiator surface is divided into six independently controlled regions, where each region consists of an array of ultra-miniature (0.5 mm x 0.3 mm) MEMS shutters fabricated by means of the Sandia National Laboratories SUMMiT-5 Polysilicon surface machining process. When a shutter is "open" it is in the high emittance state, and when a shutter is "closed" it is in the low emittance state. The emittance of each region can be independently controlled from a low value (emittance = 0.3) to a high value (emittance = 0.7) to give a dynamic range of 0.4.	TRL6	
Miniature Communications Transponder	The ST5 mission digital X-band Transponder provided uplink and downlink operation for commands and telemetry, and a coherent radio frequency turn around for Doppler tracking. It is a low mass (1.98 kg), low power (18 W), system operating with a low voltage (6.5 - 8.4 V) power supply. ST5 communications subsystem components included the transponder (4.9 W transmit and receive), a high power amplifier (1.5 W RF output), diplexer, band pass filter, and two X-band antennas with 3 dB splitter. Twenty-four voltage regulators are used to mitigate Single Event Latchup (SEL) autonomously. The regulators momentarily remove power from the section of a circuit impacted by a current rise due to latchup. All three satellites performed well during the 90-day ST5 mission. The ST5 transponder demonstrated that it is possible to use largely commercial technologies to produce a prototype miniature flight X-band communications system.	Validated (TRL7)	

	Miniature Magnetometer	The ST5 Miniature Magnetometer is a low-power, 3-axis fluxgate, research-grade, vector magnetometer system designed to provide attitude determination and to characterize spacecraft dynamics. The system consists of a sensor head with an adaptor cable for connecting the head through the spacecraft wall to an electronics box. A boom deploys the magnetometer head and maintains it in alignment with respect to the spacecraft. The instrument makes sensitive high-rate, accurate measurements of the ambient magnetic field vector throughout the entire orbit. The three ST5 spacecraft also included an autonomous magnetometer Science Event Warning (SEW) feature. When a science event is detected by the on-board algorithm, the magnetometer data sample rate is changed from the nominal 8 Hz low-rate mode to a 16 Hz hi-rate mode which remains active for 10 minutes. The magnetometer system on all three spacecraft functioned as expected. The ST5 spacecraft was validated as a suitable platform for making research quality science measurements.	Validated (TRL7)	**Solar Terrestrial Probes, Living With a Star, and Explorer missions** **Magnetospheric Multiscale Mission (MMS)** ST5 miniature magnetometer is the basis for the MMS fluxgate magnetometer
	Miniature Spinning Sun Sensor	The ST5 Miniature Spinning Sun Sensor (MSSS) consists of a two-reticle spinning Sun sensor and processing electronics packaged in one unit. The MSSS has installed stray light baffles, weighs 173 g, and draws 63 mW from a regulated 5.23 Vdc power source. The outputs of the system consist of a Sun pulse output, an eleven-bit parallel data output, and a thermistor output. The pulse occurs when the Sun passes the command plane normal to the sensor. The parallel data output is a gray coded digital word that measures the angle of the Sun line in the command plane. Data from the MSS was used in support of ground-based spacecraft attitude determination, and to assess Sun sensor performance (field of view, resolution, accuracy) by comparison with data from the magnetometer.	Validated (TRL7)	**THEMIS**
	Nutation Damper	The nutation damper carried on each of the three ST5 spacecraft passively controls spacecraft nutation. The dampers are in the form of a square-shaped tube with 19.5-cm sides, a tube outer diameter of 1.27 cm, and wall thickness of 0.14224 cm. Their weight is 236 g. The dampers are fully filled with Dow Corning Silicone 200 fluid. The nutation dampers damped out nutation introduced during initial deployment from the launch vehicle and magnetometer boom release, and following attitude precession and delta-V maneuver thruster firings. Their performance was determined based on spacecraft attitude data from the Sun sensor and the magnetometer. The nutation damper by design was tuned for an optimal less than 60 min time constant at spacecraft spin rates of 28 to 30 rpm. The damper time constant increases as the spin rate deviates from this range. Flight data results verified the required performance, and also the requirement that steady-state nutation angles be less 0.5 deg.	Validated (TRL7)	
	Spacecraft Deployment Mechanism	A Spacecraft Deployment Mechanism released each of the three ST5 spacecraft from the launch vehicle while imparting a nominal spin. The modular design mechanism supports the entire spacecraft and provides the electrical interface to the launch vehicle, including a separation signal to the launch vehicle. ST5 demonstrated the first use of a low-shock shape-memory alloy pinpuller device to initiate spacecraft deployment. The spacecraft separation connector, on a compliant mount, has high tolerance for misalignment, while allowing for separation with zero force. The deployment mechanism successfully separated each spacecraft from the launch vehicle on command on launch day. The initial spin rates and nutation angles were as predicted, < 2 degrees. The nutation damper subsequently reduced the nutation angle to << 1 degree. The spin rates measured after boom deployment were 18.60, 19.01, and 27.39 rpm for the forward, mid and aft spacecraft respectively.	Validated (TRL7)	
ST6 ASE (Mission Complete)					ST6 ASE software was uploaded to EO1 in May 2003 to be used with data from the Hyperion instrument. Technology validation was completed in May 2004. The ASE software has been in continuous operation on the EO1 extended mission since January 2005.
	Autonomous Sciencecraft Experiment (ASE)	ASE software enables new classes of missions in which the spacecraft conducts a highly interactive investigation, thus impressively increasing mission science return. This technology is applicable to a wide range of missions, including subsurface explorers, autonomous rovers, and coordinated systems of multiple spacecraft/sensors. ASE enables autonomous spacecraft response to detected science events, thus simplifying labor intensive ground based mission planning and analysis functions. The benefits enabled include: • Returning the most important data • Fast reaction time to dynamic science events • Reduced operational cost • More responsive spacecraft to unknown environments.	Validated (TRL7)	**Mars Odyssey** Science detection software proposed for extended mission to track formation and retreat of CO2 icecaps, to identify thermal anomalies, and to track dust storms. **Mars Exploration Rovers - 1** Science event detection software uploaded and operational enabling onboard detection of dust devils and clouds in MER imagery. **EO1 Extended Mission** Used in conjuction with other Earth-observing spacecraft (besides EO-1 and including Terra, Aqua and GOES) and ground sensors (primarily around volcanoes) to form an autonomous sensor-web to study volcanic activty, floods, sea-ice topography, and wild fires. **Mars Exploration Rovers - 2** ASE software adapted for AEGIS (Autonomous Exploration for Gathering Increased Science) application to enable MER rovers to perform automated, targeted remote sensing during and after traverses.
ST6 ISC (Mission Complete)					The ST6 ISC is one of eleven experimental payloads flying on the TacSat-2 mission managed by the Air Force Research Laboratory. Validation was completed in July, 2007.

	Inertial Stellar Compass (ISC)	ISC is a real-time, miniature, inertial stellar attitude determination system connecting two separate assemblies: the camero-gyro assembly and the data processor assembly. An active pixel sensor in a wide-field-of-view miniature star camera (star tracker) and microelectromechanical system (MEMS) gyros form a very low mass, low power compass unit which provides extremely accurate information for navigation and control. The star camera takes pictures of star patterns and reports them for comparison to the celestial map in the spacecraft computer. Every couple of minutes the star camera sends data to the gyroscopes to correct them for drift and bias. The gyroscopes continuously monitor the spacecraft's motion and inform the control system how to keep the spacecraft stable and pointed in the right direction. The ISC shifts operating decisions to the spacecraft itself enabling autonomous recovery of orientation and after a temporary malfunction or power loss.	Validated (TRL7)	**AFRL BAA-VS-07-03** Proposed to AFRL in January 2007. This proposal offered the DS1 Autonomous Optical Navigation technology with the ST6 ISC to demonstrate a compact package for performing tasks related to space situational awareness.
ST7 (Fab, Assy, Test & Launch Ops)					Colloidal Micro-Newton thrusters to fly on ESA LISA Pathfinder spacecraft. Spacecraft will operate in "halo" orbit about the Sun-Earth L1 point (about 1.5 million km from Earth towards Sun).
	Colloidal Micro-Newton Thrusters	Colloidal micro-newton thrusters are miniature ion engines propelled by colloidal fluid fed through a needle by a pressurizing system. A high electrical field applied to the tip of the needle causes droplets to form and be ejected from the needle. The droplets are spontaneously ionized and accelerated by high voltage, to generate variable thrust in the desired range. This new enhanced micro-thruster technology has high-stability dual electrodes, a high-stability propellant feed system, nanotube field emission, and high voltage converters. The thrusters provide spacecraft position control, keeping noise forces on freely-floating test masses small enough to meet high precision acceleration-noise performance goals. Two clusters with four micro-newton thrusters each and control software residing in a dedicated computer constitute the NASA-funded ST7 Disturbance Reduction System (DRS). Gravitational Reference Sensors (GRS) provided by the European Space Agency (ESA) are interlocked with the DRS to form a combined system to be flight validated on the joint NASA and ESA Laser Interferometer Space Antenna (LISA) Pathfinder mission in late 2009 or 2010. Together these technologies will demonstrate precision spacecraft control, validating position measurement of objects in weightlessness with 100-times greater accuracy than achieved before.	TRL6	
ST8 (Fab, Assy, Test & Launch Ops)					In light of the changes in NASA strategic planning due to reprioritization of the SMD budget, the ST8 project will redefine its implementation approach to complete and test the four experiments to bring their technologies to at least Technology Readiness Level (TRL) 6. The flight segment of the ST8 project has been cancelled.
	Dependable Multiprocessor	The Dependable Multiprocessor (DM) integrates state-of-the-art commercial of the shelf (COTS) processing components, real-time environment sensing, fault tolerant control algorithms, and rad-hard system infrastructure to provide an environmentally-adaptable high-performance, reliable on-board processing platform. ST8 will flight validate DM technology that allows much faster processing of complex science data on-board spacecraft. Field Programmable Logic Arrays (FPGAs) are used as "accelerators" or "reconfigurable co-processors" that for some types of science data analysis programs provide 100x to 1000x improvement over current standard space borne processors. The DM technology to be tested by ST8 will autonomously and adaptively configure the level of fault tolerance applied to the computer system in response to changing mission environments. The DM consist of three key technology enablers: • Real time radiation environment sensing • A computer architecture that supports adaptable configuration • A system controller mechanism to manage environment sensing and determination and development of appropriate fault tolerant configurations to optimize performance and efficiency while maintaining reliable operation.	TRL5	
	SAILMAST	SAILMAST (Scalable Architecture for the Investigation of the Load Managing Attributes of a Slender Truss) validates the load carrying attributes of versatile gossamer truss technology through correlation of flight measurements with analytical prediction. This new ultra lightweight, deployable mast technology intended for solar sail propulsion systems will also be useful for future missions that deploy telescope sunshades, solar array assemblies, large aperture optics, instrument booms, and antennas. The ST8 project will deploy and operate a 40 m mast and accurately measure its zero-g shape to validate modeling for masts up to 100 m long. SAILMAST utilizes flexible graphite fiber and the same proven structural elements as previous masts made of fiberglass, but optimizes their design for application on a gossamer spacecraft. Continuous longerons run the length of the mast which can be stowed at < 1% of the deployed length. SAILMAST is self-deployed by the stress energy in the coiled and stowed longerons. A lanyard controls the rate of deploymentt. The unit mass of the 40 m deployable boom experiment system is < 40 g/m compared to ~ 93 g/m for fiberglass masts.	TRL5	**DSX** DSX is a experimental mission sponsored by the USAF Research laboratory to be launched in late 2009 -- early 2010. Two 40-meter Sailmast booms will serve as the 80-meter long antenna for the DSX Wave Particle Interaction Experiment. **Magnetospheric Multiscale Mission (MMS)** Ten 12-meter booms ordered for the four spacecraft that will measure magnetic and fields, energetic particles, plasma waves and also report solar burst events.
	Thermal Loop	The ST8 Thermal Loop Experiment (TLE) transports large heat loads over long distances with small temperature differences and without external pumping power. It uses a single miniature loop heat pipe (MLHP) with two evaporators and two condensers, thermal electrical coolers, and two fixed radiators. A thermoelectric converter (TEC) is attached to a compensation chamber (CC) which is connected to the associated evaporator by a copper thermal strap. A flow regulator is located downstream of the condensers, and coupling blocks connect the vapor and liquid lines.	TRL6	
	UltraFlex 175	UltraFlex-175 is an ultra-lightweight, accordion-fanfold, flexible, blanket solar array that deploys into a tensioned rigid, pre-loaded structure. UltraFlex-175 provides significant advancement in power-to-mass performance over existing state-of-art high performance solar arrays based on rigid composite honeycomb panel construction. The solar array is composed of ten primary interconnected isosceles triangle shaped ultra-lightweight substrata forming a 2.9 m diameter array assembly. When deployed in a rotational fan fashion, each triangular substratum (also known as a "gore") unfolds. Upon full deployment the structure becomes tensioned into a rigid shallow umbrella-shaped structure. Radial spar elements and in-plane tensioning springs attached to the substrates elastically deflect to predetermined position to maintain the structure in a high stiffness state. The baseline ST8 flight validation success criteria for a 7 kW size solar array are:	TRL5	**Advanced Next generation UltrFlex-175 program elements under development.** Larger version of UltraFlex 175 planned for Orion Spacecraft Service Module.

		• > 175 W/kg at BOL • > 0.25 Hz deployed first mode frequency • > 30 kW/m^3 stowed specific volume		
ST9 (Flight Project Cancelled)			Due to competing budget priorities, the ST9 Project was cancelled in September, 2007. This project was to develop and validate important new capabilities for precision-guided landing and hazard avoidance in future planetary exploration missions using computer vision. Flight validation was to be accomplished via four sounding rocket missions, the first of which was successfully launched from the White Sands Missile Range on 5 April 2006.
	Terrain-Relative Guidance System (TRGS)	This technology encompasses both terrain-relative navigation (TRN) and landing hazard detection (HD). TRN estimates the position and velocity of a spacecraft relative to the landing site to enable pin-point landing with 3-sigma position knowledge error < 100 m and 3-sigma horizontal velocity knowledge error of < 20 cm/sec at touchdown. This is achieved by a sequence of estimation steps using onboard descent imagery, radar altimetry, IMU data, and prior reconnaissance imagery of the landing site (the "map"). Matching descent imagery to the map prior to entry refines S/C state knowledge prior to a braking burn or atmospheric entry. Matching additional descent imagery to the map after entry refines state knowledge again before vernier thrusting to the chosen landing site. Feature matching between successive descent images maintains precise terrain-relative velocity knowledge until touchdown. An FPGA-based vision computer performs the image processing and a Kalman filter combines measurements from the vision subsystem, IMU, and radar altimeter to estimate S/C state. HD uses onboard descent cameras to detect hazards on the scale of the lander in the last kilometer of descent. Shadow detection with one descent camera detects rocks and estimates their positions, diameters, and heights from approximately 1 km altitude. Terrain elevation mapping with an onboard stereo pair of descent cameras estimates slopes over scales of 100 m starting at approximately 500 m altitude, with the scale decreasing as altitude decreases, down to 10 m or less from altitudes of 100 m and less.	TRL4	